AIAA 99-0851

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37th AIAA Aerospace Sciences Meeting and Exhibit
January 11-14, 1999 / Reno, NV

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A 2007 MARS SAMPLE RETURN MISSION UTILIZING IN-SITU PROPELLANT PRODUCTION

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A system trade study was conducted to determine the feasibility of a 2007 Mars sample return (MSR) mission utilizing the Martian atmosphere for in-situ propellant production (ISPP). A hybrid zirconia and Sabatier/Electrolysis (S/E) process ISPP system was assumed that produced liquid oxygen and liquid methane. The emphasis of the study was threefold. First, to determine what impact the choice in mixture ratio of the oxygen/methane propellant combination used for Mars ascent has on the overall injected mass from Earth of the MSR mission elements. Second, to ascertain if the 2003/2005 "workhorse" lander being designed for MSR missions can be modified to accommodate a 2007 ISPP MSR mission. Third, to identify what parameters and technologies have a significant impact on the overall injected mass of the MSR mission elements. This paper also summarizes the current status of ISPP work funded by the NASA through the Jet Propulsion Laboratory (JPL). It was determined that the choice in mixture ratio has a moderate impact on the overall injected mass from Earth of the MSR mission elements. Although it satisfies the injected mass constraint of an affordable medium-lift launch vehicle, a 2007 ISPP MSR mission cannot be accomplished using a modified 2003/2005 "workhorse" lander due to configuration and packaging issues. Configuration, propulsion, power, and thermal control appear to be the four areas with the highest impact on the overall feasibility and injected mass for an ISPP MSR mission. Technology investment in these areas is required to make a 2007 ISPP MSR feasible.

Introduction

The California Institute of Technology's Jet Propulsion Laboratory (JPL) and the NASA have outlined a roadmap for the exploration of Mars that involves several sample return missions during the first decade of the new millenium.[1] The 2003/2005 Mars Sample Return (MSR) Project, with the objective of returning the first set of samples, has just been initiated. To transport Mars samples to Earth in the least expensive, most expeditious manner, the current plan for the 2003 and 2005 MSR missions baselines using a compact solid rocket motor based ascent vehicle that does not require a sophisticated guidance system. However, to provide a bridge to eventual human exploration of Mars, which requires in-situ propellant production (ISPP) of liquid propellants for Mars ascent, the roadmap accommodates a transition to an ISPP-based MSR mission starting in 2007 if certain conditions such as cost and technology readiness are met.

The purpose of this paper is to investigate the feasibility of an ISPP-based MSR mission for a 2007 launch. This paper should be of great value in clarifying the scale and characteristics of such a mission in order to provide a target for future ISPP-related technology developments which must be completed and demonstrated by about 2004, three years prior to launch. This paper will describe a particular ISPP-based MSR mission point design. This point design was based on a two-stage Mars ascent vehicle utilizing liquid oxygen and liquid methane as the propellants used for ascent. A parametric trade was performed on the mixture ratio of this propellant combination. The choice in mixture ratio impacts the performance of the propulsion system, the amount of liquid hydrogen that must be brought from Earth, the production rate of propellants on Mars, and the size of propellant tanks on the ascent vehicle. The choice in mixture ratio also impacts the power level required to run the ISPP system and cryogenically maintain the propellants. These effects, in turn, impact the entire size and mass of all the MSR mission elements. It was determined that the choice in mixture ratio has a moderate impact on the overall injected mass from Earth of the MSR mission elements. The lowest injected mass of the MSR mission elements occurs at a mixture ratio of 3.9. Based on the assumptions made in this paper, a 2007

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ISPP MSR mission satisfies the injected mass constraint of an affordable medium-lift launch vehicle yet cannot be accomplished using a modified 2003/2005 "workhorse" lander due to configuration and packaging issues.

Ground Rules and Major Assumptions

This section outlines the ground rules and major assumptions that defined this study. The major assumption regarding orbital rendezvous vs. direct return is discussed first. A description of the major assumptions regarding the ISPP System, Mars Ascent System (MAS), Lander, and Earth Return Vehicle (ERV) follow.

Orbital Rendezvous vs. Direct Return

Current planning for non-ISPP-based MSR missions involves use of "dumb" 3-stage solid ascent rockets that provide the propulsive capability to launch a spherical sample container into a low-Mars orbit. This orbiting sample (OS) would have a relatively high reflectivity and contain a few hundred grams of Martian rocks and soil in a sealed sample container. A solar-powered beacon would also be included in the OS to further aid retrieval by an Earth Return Vehicle (ERV) waiting in orbit. This has the great advantage that the ERV need not be lifted from the surface of Mars, thus greatly reducing the propellant requirement compared to a direct return to Earth mission in which the ERV would have to be lifted from the surface of Mars. Furthermore, use of a "dumb" ascent vehicle lightens the ascent vehicle considerably by eliminating sophisticated guidance electronics.

Even though the orbit into which such a "dumb" ascent vehicle will place the OS is expected to be known only to within ~1° in inclination and ~100 km in semi-major axis, the ERV will be able to find and capture the OS using available technology. This approach also has the programmatic advantage of breaking the mission into separate launches of the ERV and Mars Ascent System/Lander, using moderate-cost launch vehicles. If these two launches occur in different fiscal years, it has the effect of spreading the costs out over time, which is advantageous to the Mars Exploration Program which has level funding. The disadvantages of this approach are (i) that multiple launches are costly, and even if yearly costs are spread out, the total cost might be relatively high, and (ii) the use of orbital rendezvous and multiple vehicles with sample transfer may be risky.

Direct return from the surface of Mars to Earth would eliminate the risky time-consuming steps of finding and capturing the OS, but would complicate planetary protection efforts. For a "bring your own propellant" (BYOP) mission, the amounts of propellant required by the ascent vehicle for direct return would be so large that no credible launch vehicle would be adequate. However, for an ISPP-based MSR direct-return mission, the large amounts of propellant required could be produced on Mars and this might make it possible to carry out such a mission on a very large launch vehicle. Nevertheless, preliminary rough calculations indicate that even with ISPP, and using a heavy-lift launch vehicle, mass margins are likely to be very tight, and volumetric constraints will be difficult to overcome.

By contrast, if an ISPP-based MSR mission is designed to emulate as much as possible the first two MSR missions (which utilize "dumb" solid ascent rockets), the considerable technology and engineering work developed for these two MSR missions would be transferable to the ISPP-based MSR mission. This use of developed and flight-proven heritage (by 2007) would greatly reduce the cost of the ISPP-based mission. That is to say, the "dumb" solid rocket-based ascent vehicle used in 2003 and 2005 could be replaced in 2007 by a "dumb" liquid-fueled ascent vehicle with propellants produced by ISPP.

The physical size of this 2007 "dumb" liquid-fueled ascent vehicle would be very small compared to the scale of an ISPP-based human ascent vehicle. This approach has advantages and disadvantages. The major advantage is that a successful end-to-end ISPP system and associated ascent vehicle would validate the technology and concept of using ISPP. Another advantage is that the ISPP-based ascent vehicle would be sized to use the existing sample container developed for the 2003 and 2005 ascent vehicles. Using an identically sized sample container eliminates the need to redesign elements of the ERV that are used to find, capture, and return the sample to Earth. A disadvantage is that the scale of the ISPP system would be quite small compared to the ISPP systems that would be used to lift humans off the surface of Mars. Since the use of ISPP is viewed as a "bridge" to eventual human exploration of Mars, the scale-up from this small demonstration to a full-size ISPP system for humans would be several orders of magnitude larger.

Nevertheless, this paper is dedicated to the ISPP mission with a "dumb" ascent vehicle that provides the propulsive capability to launch a spherical sample container into a low-Mars orbit because it appears eminently more technically and financially feasible than a much larger scale direct-return ISPP MSR mission.

ISPP System

The ISPP system is used to produce the propellants required by the MAS for ascent from the Martian surface. This is done in three steps: Mars atmosphere acquisition and compression; conversion of pressurized carbon dioxide to propellants in a reactor; and liquefaction and storage of propellants. A hybrid ISPP system was assumed using both the zirconia and Sabatier/Electrolysis (S/E) processes. The zirconia system produces only oxygen while the S/E system produces both oxygen and methane. Hydrogen required by the S/E process is brought from Earth in an insulated cryogenically cooled propellant tank. A 300 day production time was assumed. The ISPP System is discussed in detail in a subsequent section.

Mars Ascent System (MAS)

The Mars Ascent System (MAS) provides the propulsive capability to launch a sample container into a low-Mars orbit for later retrieval by the ERV. A two-stage to low-Mars orbit MAS was assumed in this study. The basic configuration of the MAS assumed in this study is illustrated in Figure 1.

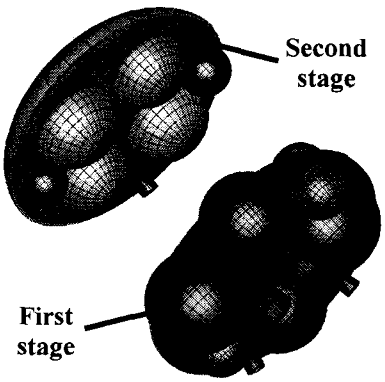

Figure 1. MAS Configuration Assumed in this Study.

Both stages assume liquid oxygen and liquid methane as the propellants used for ascent. The propulsion system assumed a warm-gas pressurization system, lightweight composite-overwrapped tanks, and lightweight components. The pseudo-guided MAS utilizes a lightweight avionics package that is simple enough to achieve a low-Mars orbit. Once the MAS achieves a low-Mars orbit it ejects the sample container and "dies". The MAS is described in depth in a subsequent section.

Lander

The Lander is comprised of a cruise stage, backshell, heatshield, and landed element. The Lander utilizes the heatshield, parachutes, and a propulsion system to soft-land the MAS on the surface of Mars. The landed element provides direct-to-Earth communication link, power to the ISPP system, and supports the MAS until it launches. The basic configuration of the Lander assumed in this study is illustrated in Figure 2.

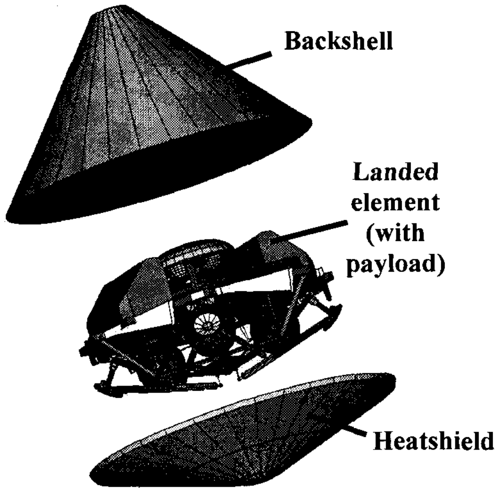

Figure 2. Lander Configuration Assumed in this Study (Cruise Stage not shown).

The MAS, ground support equipment, and elements of the ISPP system are illustrated on the deck of the Lander in Figure 2. Certain items such as descent thrusters and tanks are below the deck and not visible in Figure 2. The lander is discussed in detail in a subsequent section.

Earth Return Vehicle (ERV)

The purpose of the ERV is twofold. First, the ERV provides the primary propulsive capability on the way to Mars, at Mars, and on the way to Earth. Second, the ERV must find, rendezvous, and capture the OS. The captured OS is placed in an Earth entry vehicle (EEV) carried by the ERV. The EEV provides a sealed environment in which the sample can survive Earth re-entry and landing. A rendering of the ERV and EEV is illustrated in Figure 3.

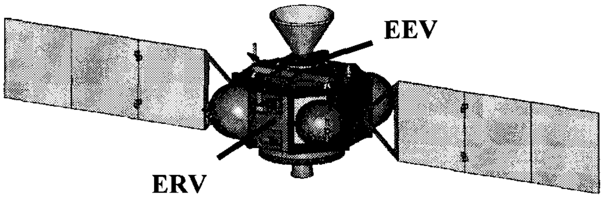

Figure 3. Potential ERV and EEV Configuration.

The Mars Exploration Program

In the wake of the loss of the Mars Observer spacecraft in 1993, NASA and JPL conceived a new approach to the exploration of the planet Mars based on frequent, lower-cost missions instead of larger, more expensive, and necessarily less frequent Observer class missions.[2] The Mars Surveyor Program (MSP) that emerged involves sending at least one spacecraft to Mars at every opportunity (roughly every 26 months).[1] Furthermore, the MSP is to be accomplished under a level funding profile. The first of these missions launched was the Mars Global Surveyor, which is currently in orbit around Mars and which will begin mapping the Martian surface in March 1999. The next set of missions is the Mars Climate Orbiter and Mars Polar Lander, launched in December 1998 and January 1999, respectively. These missions are to be followed in 2001 with copies of these two engineering systems with the minimum modifications required to accommodate a different suite of scientific and engineering experiments, and to allow a different landing latitude for the lander.

The aforementioned missions will enhance our knowledge of Mars and our engineering tools to allow the launch of the first MSR mission beginning in 2003. This system, as currently envisioned, will feature a much larger lander capable of carrying a compact solid rocket motor based Mars Ascent Vehicle (MAV), extensively instrumented rover, and possibly other scientific experiments. The rover will collect soil and rock samples from the surrounding area, transferring them to a sample container onboard the MAV. When an appropriate number of samples have been collected, the MAV launches and caches the sample container in a low-Mars orbit for future retrieval. This 2003 lander may be complimented by the orbital observations by the European Space Agency/Italian Space Agency's (ESA/ASI) Mars Express Orbiter.

The baseline plan for the 2005 MSR mission will fly copies of the 2003 lander, MAV, and rover as well as the first flight of an Earth Return Vehicle (ERV) provided by the French space agency CNES. The ERV contains a Earth Entry Vehicle (EEV) provided by NASA. The 2005 rover will collect soil and rock samples in a similar manner as the 2003 rover, transferring them to a sample container onboard the MAV. When an appropriate number of samples have been collected, the 2005 MAV launches and caches the sample container in a low-Mars orbit. The CNES ERV collects one or both of the 2003 and 2005 orbiting samples and returns them to Earth via the EEV in 2008.

Possible 2007/2009 Mars Sample Return Mission Profile

The 2003 and 2005 MSR missions provide a logical progression in the development of a potential ISPP-based MSR mission in 2007. This system will feature a copy of the 2003/2005 lander with only the minimum modifications required to accommodate an ISPP system, ISPP-based MAS, robotic arm, and possibly other scientific experiments. The following section describes a possible mission profile for the 2007 and 2009 ISPP-based MSR mission described in this paper.

September 14, 2007

The 2007 MSR mission begins with the launch of a Boeing Delta IV Medium or Lockheed-Martin EELV MLV-class launch vehicle from Kennedy Space Center in Florida on a Type II trajectory to Mars. Within the launch vehicle payload shroud is the MAS, Lander, and ISPP system. These three elements are encased in an aeroshell. A cruise stage is attached to the aeroshell to provide power, telemetry, communication, and propulsion during the 50-week trip to Mars.

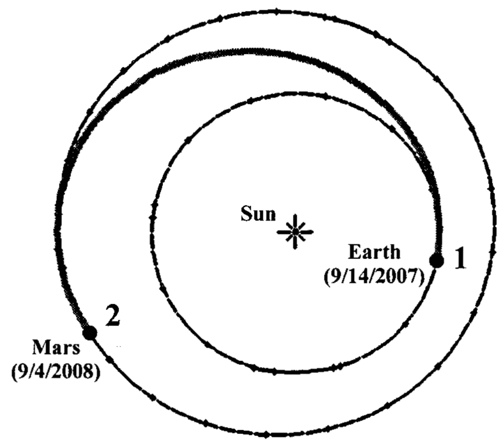

Figure 4. Trajectory Profile of a Potential 2007 Mars Sample Return Mission.

September 4, 2008

When the aeroshell and cruise stage arrive in the vicinity of Mars, the aeroshell (with the MAS, Lander, and ISPP system inside) separates from the cruise stage and enters the Martian atmosphere directly at several kilometers per second. The aeroshell absorbs the heat generated by the atmospheric entry, slowing the aeroshell "package" down to a few hundred meters per second. The aeroshell is discarded and parachutes are deployed from the Lander to further slow the descent. The Lander touches down at a latitude of ~15° N. This latitude was selected to provide a balance between scientific interest and power generation over a 300 day span. The Lander and ISPP system deploy solar arrays immediately to obtain power from the Sun. Sample collection via a robotic arm and ISPP production begins as soon as possible.

July 1, 2009

After approximately 300 days in which Martian rock, soil, and atmospheric samples have been obtained and the propellants required by the MAS for ascent have been produced, the MAS is readied for launch. The propellant loading tubes joining the ISPP system to the MAS are disconnected and the MAS launches into a ~600 by ~600 km orbit. Once the MAS has achieved the desired orbit, the sample container is ejected and the MAS "dies". The orbiting sample (OS) is now cached in a low-Mars orbit for future retrieval by an ERV.

October 7, 2009

The 2009 MSR mission begins with the launch of two Boeing Delta IV Medium or two Lockheed-Martin EELV MLV-class launch vehicles from the Kennedy Space Center in Florida on a Type II trajectory to Mars. Within the first launch vehicle payload shroud are copies of the 2007 MAS, Lander, and ISPP systems. As in the 2007 MSR mission, these three elements are encased in an aeroshell with an attached cruise stage. Within the second launch vehicle payload shroud is an ERV carrying an EEV. It should be noted that these two sets of payloads could instead be launched together on a Delta IV Heavy, Lockheed-Martin EELV HLV, or Arianespace Ariane 5-class launch vehicle.

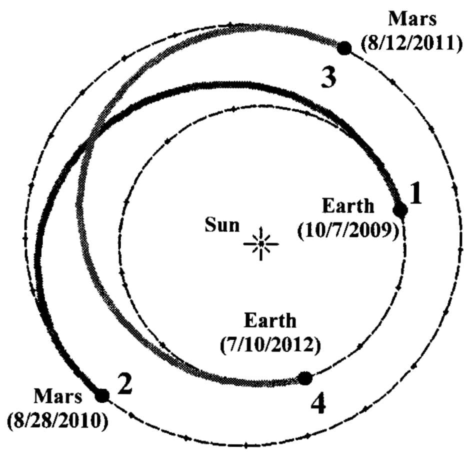

Figure 5. Trajectory Profile of a Potential 2009 Mars Sample Return Mission.

August 28, 2010

Forty-six weeks after launch, the two MSR elements arrive in the vicinity of Mars. The ERV enters either a highly elliptical Mars orbit via aerobraking or a low-Mars orbit via aerocapture. Meanwhile, the aeroshell (with the MAS, Lander, and ISPP system inside) separates from the cruise stage and enters the Martian atmosphere directly. The Lander effects a Mars entry in a similar manner to its 2007 counterpart. The 2009 Lander touches down a latitude of ~15° N. This latitude was also selected to provide a balance between scientific interest and power generation over a 300 day span. The Lander and ISPP system deploy solar arrays immediately to obtain power from the Sun. Sample collection via a robotic arm and ISPP production begins as soon as possible.

June 24, 2011

Approximately 300 days pass in which Martian rock, soil, and atmospheric samples have been obtained and the propellants required by the MAS for ascent have been produced. During this time, the ERV tracks and captures the cached 2007 OS. When the MAS is readied for launch the propellant loading tubes joining the ISPP system to the MAS are disconnected and the MAS launches into an low-Mars orbit. Once the MAS has achieved the desired orbit, the sample container is ejected and the MAS "dies". The ERV tracks and captures the 2009 OS over the next two weeks. The 2007 and 2009 OS are both stored in a single Earth Entry Vehicle (EEV) that is attached to the ERV.

August 12, 2011

With two OSs captured, the ERV injects back to Earth on a Type II trajectory.

July 10, 2012

Forty-seven weeks after leaving a low-Mars orbit, the ERV arrives in the vicinity of Earth. Given that the containment of the samples has been validated, the EEV is spin ejected and enters the Earth's atmosphere on a trajectory targeted for the Utah Test and Tracking Range (UTTR). The EEV is collected and brought to a quarantined facility where the samples brought from Mars are thoroughly investigated. In the event the containment of the samples can either not be verified or has been breached, the mission will be aborted and the ERV/EEV will be deflected away from Earth.

The characteristics of the 2007 and 2009 MSR trajectory opportunities described in this section are summarized in Table 1.

Table 1. Characteristics of the 2007 and 2009 MSR Trajectory Opportunities.

Trajectory	Launch	Arrival	Notes
Earth–Mars	9/14/2007	9/4/2008	C_3 = 13.25 km^2/sec^2
Earth–Mars	10/7/2009	8/28/2010	C_3 = 10.70 km^2/sec^2
Mars–Earth	8/12/2011	7/10/2012	349 day stay time @ Mars
Mars–Earth*	9/28/2013	8/29/2014	1127 day stay time @ Mars

*indicates back-up trajectory if primary (8/12/2011) return trajectory is missed

It is apparent from Table 1 that the driving hyperbolic excess requirement of the launch vehicle (C_3) is the 2007 launch (13.25 km^2/sec^2). It should also be noted from Table 1 that a back-up trajectory exists for the return of the ERV to Earth. This could happen if either ISPP production takes significantly longer than 300 days or tracking/capturing the 2007 and/or 2009 OS takes longer than anticipated. The drawback of the back-up trajectory is the two-year delay in the return of the Martian samples.

In-Situ Propellant Production Options and Assumptions

An ISPP system for a MSR mission is a chemical processing plant, complete with its own cryogenic storage system, which must function remotely and autonomously for typically several hundred days on the surface of Mars. The need to operate for such long periods is dictated by the fact that only a relatively small, low-mass ISPP production plant can be landed on Mars, but if it operates for a sufficiently long period, it can produce many times its own weight of propellant. Because of the nature of the orbits of Earth and Mars, it is propitious to return from Mars approximately two years after launching from Earth. This allows hundreds of days of ISPP operation on the surface of Mars prior to return.

The elements of the ISPP process assumed are

- Mars atmosphere acquisition and compression
- conversion of pressurized carbon dioxide to propellants in a reactor
- liquefaction and storage of propellants
- hydrogen storage

This section describes these processes in detail.

Mars Atmosphere Acquisition and Compression

The atmosphere of Mars is ~95.5% carbon dioxide (CO_2), with the remainder made up mainly of argon (Ar) and nitrogen (N_2), and smaller amounts of carbon monoxide (CO) and oxygen (O_2). The carbon dioxide is the feedstock to produce oxygen and possibly hydrocarbon production if hydrogen is available. Atmospheric pressure varies with hour and season but a rough average value is ~800 Pa (~6 torr). It is desirable to compress the carbon dioxide to a pressure of typically ~100,000 Pa (~1 bar) in order to facilitate conversion to propellants in a small reactor without excessive volumetric flow rates.

Thus the first step in the process is acquisition and compression of the Mars atmosphere. The leading candidate for this step is a sorption compressor. A Mars atmosphere acquisition and compression (MAAC) unit is cooled at night by radiation to the night sky while open to the atmosphere to adsorb carbon dioxide on a sorbent material. The adsorbed carbon dioxide is released at high pressure during the day by heating the MAAC. If a selective surface (high solar absorptivity, moderate emissivity) is used as the upper surface of a relatively flat rectangular sorption bed, much of the required power during the day can be acquired by direct solar heating.

Although the theory of such a device is understood to some extent, the quantitative performance of a realistic unit has yet to be demonstrated in a Mars gas mixture. Sorption compressors have been demonstrated to perform very well over a diurnal cycle when exposed to pure carbon dioxide. However, problems developed when they were tested with a Mars atmospheric gas mixture.[3,4] As carbon dioxide is extracted by the sorbent material, it is believed that the less-adsorbed gases (argon and nitrogen) tend to build up significant concentrations around the sorbent material, thus creating a diffusive barrier to further adsorption of carbon dioxide. Because of this, the residual gases within the sorbent bed need to be periodically flushed from the system to enhance continuous carbon dioxide adsorption. One approach is to periodically pass a slug of high-pressure carbon dioxide (collected and stored in a small tank during the previous day) through the sorption bed to purge these gases. P. Karlmann is presently testing this approach at the Jet Propulsion Laboratory (JPL) in Pasadena, California. This paper will assume that this approach is effective, even though it has yet to be demonstrated.

It is worth noting that the AlliedSignal company (Morristown, NJ), partly with JPL funding, recently developed a carbon molecular sieve (CMS) sorbent material which they claim provides roughly 0.3 kg of carbon dioxide per kg of CMS around a cycle from 200 K to 430 K.

Conversion of Pressurized Carbon Dioxide to Propellants in a Reactor

Although several possible candidate processes can be conjectured for ISPP on Mars, we have selected a specific approach that utilizes the most mature technologies available today. The two most mature processes are zirconia solid state electrolysis and the Sabatier/electrolysis (S/E) process.[5] The zirconia process is not quite as mature as the S/E process and is limited to oxygen production only. The S/E process has been demonstrated to be a viable process for producing both methane and oxygen. This paper assumes that both the zirconia and the S/E processes are used, each producing roughly half the amount of oxygen required, with the methane produced by the S/E process acting as the fuel. The balance between the S/E and zirconia processes can be adjusted to any desired liquid oxygen/methane propellant mixture ratio.

Zirconia Solid State Electrolysis

In the zirconia process, a solid state yttria stabilized zirconia (YSZ) ion conductor is used, which unlike a typical metal which conducts electrons, conducts electricity by means of negatively charged ions. The doped crystal lattice contains "holes" allowing oxygen ions to move through the lattice when an electric field is applied across it. Mounting metallic electrodes on each side of the zirconia and applying a difference in potential generates an electric field. In a zirconia cell, hot carbon dioxide is brought into contact with a catalyst on the cathode, thus causing some dissociation. Oxygen atoms in contact with the cathode pick up electrons to form oxygen ions (O^{--}) which are transported through the zirconia to form pure oxygen on the other side at the anode. The basic equation for this process is

$$2CO_2 + electricity \xrightarrow{zirconia} 2CO + O_2 \qquad (1)$$

It is apparent from equation (1) that two moles of carbon dioxide produce one mole of oxygen.

Thermal cycling poses significant challenges to zirconia stacks. Each afternoon as electrical power availability diminishes, the stack will have to be shut down. In the morning, it needs to be pre-heated as rapidly as possible before gas is introduced.

A number of zirconia devices have been built and tested over the years. Initially, zirconia tubes were used, but it was realized that only a stack of zirconia sheets connected in series would suffice for Mars applications. Several investigators have been successful in operating a single flat disk of zirconia, but developing a workable stack of disks has been more difficult. Recently, AlliedSignal (under JPL funding) has demonstrated successful operation of a stack of three wafers, which provides optimism that larger stacks are probably feasible. This is a significant breakthrough for zirconia technology, and appears to open the possibility of full-scale zirconia stacks for Mars application.[6,7]

Using data acquired from test of the stack of 3 wafers by AlliedSignal, we make the following estimates. Experimental data indicate that a zirconia cell stack can operate at around 0.4 amps/cm^2 with about 1.7 V across each zirconia wafer at 1073 K (800 °C). Conversion of 80% of the carbon dioxide to carbon monoxide has been demonstrated and 90% is believed to be feasible. The analysis in this paper assumes 90% conversion efficiency.

A basic quantity that relates the current in amperes (A) to the oxygen gas flow rate is:

$$1\ A = 3.79\ sccm\ of\ O_2 = 0.325\ g/hr\ of\ O_2$$

The power (in watts) required to generate an oxygen production rate of 0.325 g/hr is therefore simply the voltage across the cell. Assuming a production rate of 0.4 kg of oxygen per 7 hours requires a production rate of 57 g/hr of oxygen. This requires 176 A, or a power level of ~300 W (176 A × 1.7 V). At a current density of 0.4 A/cm^2 this corresponds to a zirconia wafer area of 440 cm^2, which might imply a stack of six 10 cm diameter wafers. Additional power inputs may be required to pre-heat incoming carbon dioxide and to compensate for heat losses from the zirconia stack. Rough estimates of heat loss from an insulated zirconia stack indicate that perhaps 60 W of additional heat input may be needed to compensate for thermal losses. Therefore the total power input to the zirconia stack is estimated at 360 W.

For a zirconia system operating at 1073 K (800 °C) and 90% conversion that produces 57 g of oxygen per hour, the input carbon dioxide flow rate is 175 g/hr, and the spent carbon monoxide/carbon dioxide stream would be 120 g/hr. The sorption compressor must supply 175 g/hr of carbon dioxide for 7 hours per day. The performance of a sorption compressor was modeled with a sky-facing radiator with a selective surface. Taking into account the solar heat gain to the radiator during the day (average of ~60 W), the average electric power level to operate the sorption compressor during the day is around 45 W neglecting thermal losses. When thermal losses are included, it is likely that the total electrical power for the sorption compressor will average out to perhaps 60 W during the day.

The total power required by the zirconia system is therefore 420 W. This does not include power required for the oxygen cryocooler. The mass of the zirconia sorption compressor is estimated using equation (1) and the assumptions provided:

$$m_{sorbent} = (0.4\ kg\ O_2)\left(\frac{1\ mol\ O_2}{32\ kg\ O_2}\right)\left(\frac{2\ mol\ CO_2}{1\ mol\ O_2}\right)$$
$$\left(\frac{44\ kg\ CO_2}{1\ mol\ CO_2}\right)\left(\frac{3.5\ kg\ sorb.}{1\ kg\ CO_2}\right)\left(\frac{1.00}{0.90}\ eff.\right)$$
$$m_{sorbent} = 4.3\ kg$$

The mass of the sorbent compressor housing and connections is estimated to be 40% of the total sorbent mass (0.4 × 4.3 kg = 1.7 kg).

The zirconia system mass (including connecting tubes, valves, sensors, and controls) based on 440 cm^2 area is estimated to be 5.4 kg.

Hence, the total mass of a zirconia system (sorbent, sorption compressor, tubes, valves, etc.) to produce 0.4 kg of oxygen per 7 hour operational Martian day is 11.4 kg. The mass and power calculated in the aforementioned analysis are directly scalable to higher or lower production rates.

Sabatier/Electrolysis (S/E) Process

In the Sabatier/Electrolysis (S/E) process, hydrogen (brought from Earth) is first reacted with compressed carbon dioxide in a heated chemical reactor:

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O \qquad (2)$$

The reactor is simply a tube filled with catalyst. Since the reaction is quite exothermic, considerable waste heat is available for heating the MAAC sorbent bed, although most of this waste heat is in the form of heat of condensation of water, which may not necessarily be simple to recover. It is likely that the hot gases exiting the Sabatier reactor could furnish some of the heat required to operate the sorbent bed.

The methane/water mixture is separated in a condenser, and the methane is dried, and stored for use as a propellant. The water is collected, deionized, and electrolyzed in a standard electrolysis cell:

$$2H_2O + electricity \rightarrow 2H_2 + O_2 \qquad (3)$$

The oxygen is stored for use as the oxidizer and the hydrogen is recirculated to the chemical reactor. Note from equations (2) and (3) that only half as much hydrogen is produced as is needed for reaction, illustrating that an external source of hydrogen is necessary for this process to work.

The S/E process was studied in some detail using a breadboard demonstration unit that worked very effectively.[8] The great advantage of this process is that it is well understood and seems to perform with high conversion efficiency, good energetics, and reliable start-up and shut down capabilities. The breadboard Sabatier reactor built was small, lightweight, and required no power for continuous operation. An external electric heater provided startup power for pre-heat prior to operations. A highly active catalyst enabled self-sustaining operation while producing chemical conversion efficiencies of over 99% for both reactants when operating with the hydrogen recovery pump. Separation of the water into hydrogen and oxygen is a well-developed process. The Hamilton-Standard Company (Windsor Locks, CT) supplied the electrolysis unit that was used for these experiments. This unit was a highly efficient and durable device based on space and military membrane and catalyst technology. A hydrogen membrane pump recovered nearly all of the residual hydrogen gas in the methane stream for re-use by the reactor. The value of the pump was enhanced by the desire to operate the Sabatier reactor with an excess of hydrogen in order to consume almost entirely the carbon dioxide. This excess hydrogen could easily be removed from the product stream with the pump. Demonstration runs exceeded 99% efficiency for both reactants.

There are two closely coupled problems in the use of the S/E process. The primary problem is the necessity to bring hydrogen from Earth in order to carry out the process (although there is also the possibility of obtaining water indigenous to Mars, it was not assumed in this paper). Closely related to this problem is that the S/E process produces an excess of methane compared to the amount of oxygen produced, which in turn, requires extra hydrogen if the excess methane is vented. Using the zirconia process to produce enough oxygen to react with the excess methane from the S/E process can minimize the hydrogen requirement.

The main power input to the S/E system is to the electrolysis cell. This requires 1.8 V_{DC}, resulting in a power requirement of 1.8 W per 0.325 g/hr of oxygen produced.[9] To produce 0.4 kg of oxygen and 0.2 kg methane per 7 hours requires a production rate of 57 g/hr of oxygen and 28.5 g/hr of methane. If this system operates for 300 days, it will produce 120 kg of oxygen and 60 kg of methane (in addition to another 120 kg of oxygen produced by the zirconia system). This requires delivery of at least 15 kg of hydrogen to Mars if there were no losses.

This paper will assume that the S/E reactor converts 100% of carbon dioxide to methane using excess hydrogen, with excess hydrogen being recovered from product stream with a membrane. The power requirement is ~315 W for the electrolysis step. The sorption compressor for the S/E system only needs to supply 80 g/hr of carbon dioxide assuming 100% conversion of the carbon dioxide. The heat requirements for this small compressor are modest and considering the availability of some waste heat from the Sabatier reactor, the electrical power needed will average less than 40 W. The total power required is then ~355 W. This does not include power required for the methane cryocooler.

The mass of the S/E sorbent is estimated roughly as follows:

$$m_{sorbent} = (0.4\ kg\ O_2)\left(\frac{1\ mol\ O_2}{32\ kg\ O_2}\right)\left(\frac{1\ mol\ CO_2}{1\ mol\ O_2}\right)$$
$$\left(\frac{44\ kg\ CO_2}{1\ mol\ CO_2}\right)\left(\frac{3.5\ kg\ sorb.}{1\ kg\ CO_2}\right)\left(\frac{1.00}{1.00}\ eff.\right)$$
$$m_{sorbent} = 1.9\ kg$$

The mass of the sorbent compressor housing and connections is estimated to be 40% of the total sorbent mass (0.4 × 1.9 kg = 0.8 kg). The S/E system mass (including connecting tubes, valves, sensors, and controls) is estimated to be 5.7 kg.

Hence, the total mass of a S/E system (sorbent, sorption compressor, tubes, valves, etc.) to produce 0.4 kg of oxygen and 0.2 kg of methane per Martian day is 8.4 kg. The mass and power calculated in the aforementioned analysis are directly scalable to higher or lower production rates.

Liquefaction and Storage of the Propellants

In-situ produced liquid oxygen and liquid methane are stored in their respective propellant tanks on the MAS. Thermal calculations indicated that 85.4 mm of insulation thickness is required on all propellant tanks to minimize the boil-off of propellant during ISPP. The mass per unit of surface area of this insulation was assumed to be 2.8 kg/m^2. Since no credible method of jettisoning the insulation was determined without significantly sacrificing the thermal insulating ability of the insulation, the insulation mass is kept during ascent. An additional 50 W/m^2 of power input per exterior tank surface area is required to maintain the in-situ produced propellants. A cryogenic cooler having a mass per unit of exterior tank surface area of 6.1 kg/m^2 was assumed to provide this power input. A 30% contingency was placed on the amount of oxygen and methane to be produced to account for boil-off.

Hydrogen Storage

There are three conceptual approaches under consideration for transporting and storing hydrogen on Mars. The first approach is adsorption on a substrate, the second is cryogenic, and the third is transporting hydrogen via a hydrogen-containing chemical compound. Adsorption of hydrogen on a substrate does not appear viable because the required mass of substrate is too large. Cryogenic storage of hydrogen can be attempted in several ways. It can be stored in the propellant tanks of the ascent system, but this appears to lead to impractical volume and mass requirements. Storage of hydrogen in a dedicated tank or tanks on the Mars Lander is a better approach but may still lead to configurational difficulties due to the large size of tankage needed. Transporting hydrogen as a hydrogen-containing chemical compound stored in the methane tanks of the ascent vehicle could be feasible if such a compound could be converted to hydrogen with a high conversion efficiency. However, such a process would add significantly to the power requirement. The only feasible approach at this time (and the one assumed in this paper) appears to be cryogenic storage in dedicated tanks on the Lander.

The mass of the hydrogen tank, insulation, and associated advanced/optimized cryogenic cooling device assumed as a function of the hydrogen mass transported is plotted in Figure 6. Figure 6 also provides the power required by the cryogenic cooling device to maintain the liquid hydrogen as a function of hydrogen mass transported.

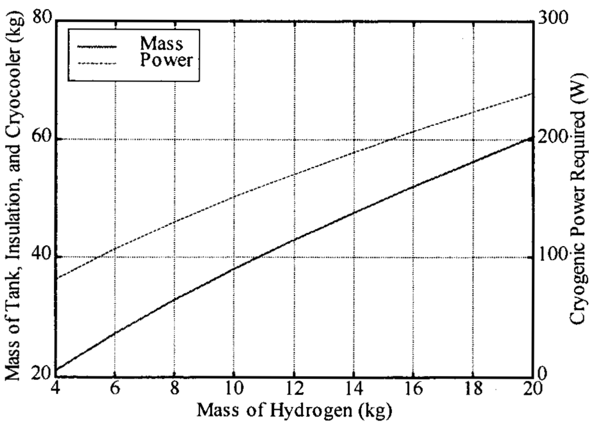

Figure 6. Hydrogen Tank Mass and Cryogenic Power Required as a Function of Hydrogen Mass.

Figure 6 illustrates the dramatic impact on mass that transporting hydrogen has as well as power that the lander cruise stage/landed element solar arrays must provide the cryogenic cooling system. A 30% contingency was placed on the amount of hydrogen required by the ISPP system.

Mars Ascent System (MAS) Assumptions

As stated in the introduction, a two-stage to low-Mars orbit MAS was assumed in this study. The total free-space equivalent delta-V required by the MAS was estimated to be 4630 m/s. The delta-V split between the first and second stage was assumed to be 2020 and 2610 m/s, respectively. This section will discuss the MAS assumptions concerning configuration, propulsion system, and other subsystems.

Configuration

The basic reference shape for the MAS is dictated primarily by packaging and volumetric constraints imposed on the Lander by the aeroshell. Both the first and second stages assume liquid oxygen and liquid methane as the propellants used for ascent. The first stage assumes two (approximately 1400 N) main engines and four 150 N thrust vector control (TVC) thrusters. The four 150 N thrusters are canted at 25° to provide adequate thrust vector control during ascent. The total average thrust of the first stage was sized to provide an initial thrust-to-Mars weight ratio of ~2.5. Preliminary trajectory calculations indicated that this initial thrust-to-Mars weight ratio was optimal for ascent. Two oxidizer tanks, two fuel tanks, and two pressurant tanks (one for the oxidizer and one for the fuel) were assumed. The second stage assumes four 150 N thrusters canted at 5° to provide both primary propulsion and thrust vector control. Two oxidizer tanks, two fuel tanks, and two pressurant tanks (one for the oxidizer and one for the fuel) were assumed. The second stage also has an aerodynamic fairing, a 29 cm × 26 cm × 12 cm box to house the avionics, a 17 cm diameter spherical sample container, and a sample container jettison device.

The complete configuration is a short, spherically blunted cylinder. The second stage shape is driven by the spherical aeroshell fairing which is required to protect the tankage, payload, and hardware from the stagnation pressures and temperatures during the high Mach number portion of the ascent. The configuration of the MAS is shown earlier in Figure 1.

Propulsion System

Propellants produced by the ISPP system are stored in their respective tanks at 83.15 K (–190 °C) and 344.7 kPa (50 psi). The first stage propulsion schematic is illustrated in Figure 7.

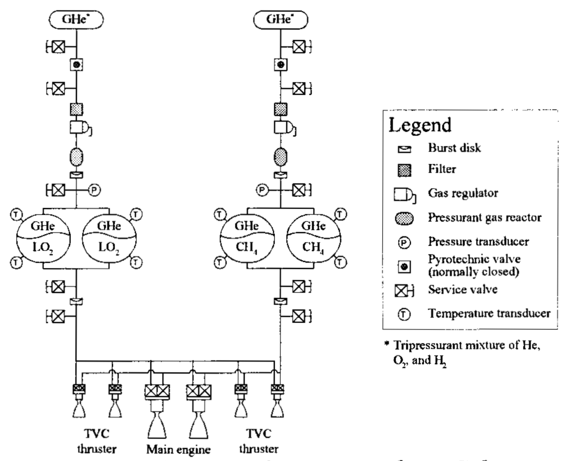

Figure 7. MAS First Stage Propulsion Schematic.

The system is activated shortly before lift-off by firing the two normally closed pyrotechnic valves downstream of the helium pressurant tanks. When the upstream pressure approaches 2.068 MPa (300 psi) regulated tank pressure, the first set of burst disks rupture, allowing the propellant tanks to pressurize. When the tank pressures approach the regulated pressure, the second set of burst disks rupture, priming the (previously evacuated) propellant lines. The propulsion system is ready to fire once the propellant lines have been primed. Service valves are provided where necessary for servicing or functional testing of the system. The service valves directly downstream of the propellant tanks are used for loading in-situ produced propellant.

The second stage propulsion system operates in a similar manner to the first stage propulsion system with the notable difference being the lack of main engines. The second stage propulsion schematic is illustrated in Figure 8.

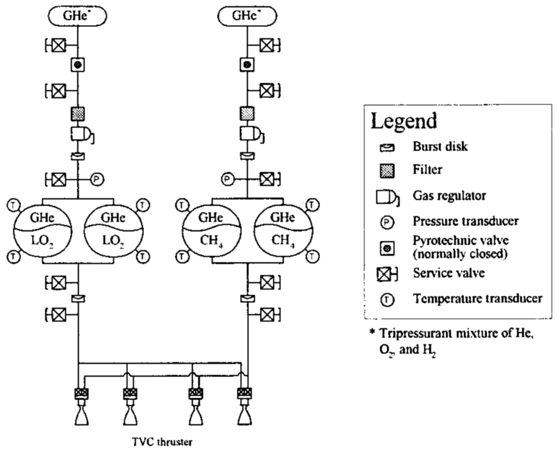

Figure 8. MAS Second Stage Propulsion Schematic.

The following subsections describe the engine mass and performance; warm-gas pressurization system assumptions; propellant and pressurant tank assumptions; and propulsion component assumptions. The development for warm-gas pressurization; propellant and pressurant tanks; and propulsion components is currently ongoing at JPL.[10]

Engine Mass and Performance

The mass of the engines/thrusters assumed in this study was based upon data provided by Jim Glass of Boeing Rocketdyne (Canoga Park, CA). Figure 9 plots this data and compares it to data provided by Jerry Sanders of Johnson Space Center/NASA (Houston, TX).

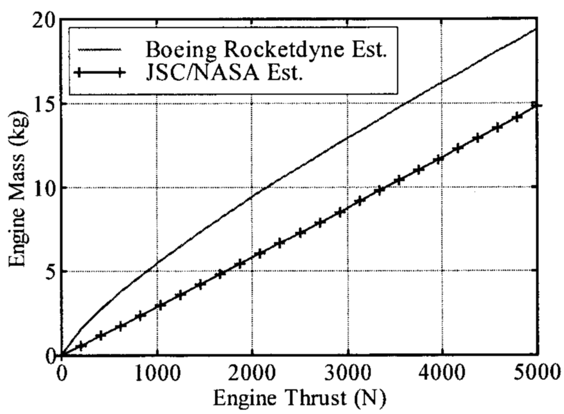

Figure 9. Engine Mass as a Function of Thrust.

Using the rigorous JANNAF procedure and assuming a parabolic wall nozzle, the theoretical specific impulse of the oxygen methane propellant combination was determined for a range of mixture ratios and thrust levels.[11] The theoretical performance (including kinetic, two-dimensional, and boundary layer loses) were reduced 2% to account for an assumed 98% combustion efficiency (i.e., vaporization and mixing efficiency). The results are plotted in Figure 10.

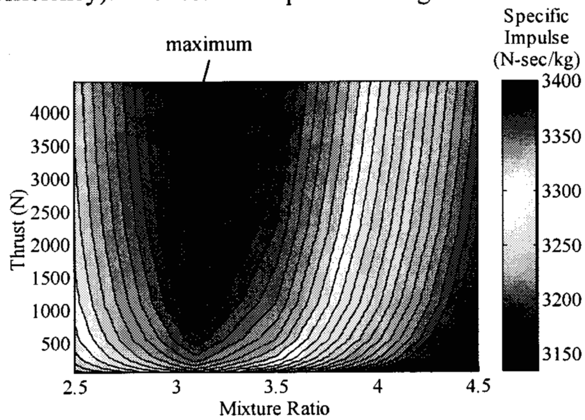

Figure 10. Specific Impulse as a Function of Mixture Ratio and Thrust.

It should be noted that no effort was made to optimize the nozzle design and there may be practical limitations imposed by chamber cooling and/or two-phase flow effects that might prevent the performances computed from actually being attained in a practical rocket design. It is apparent from Figure 10 that the specific impulse is a complex nonlinear function of both the mixture ratio and thrust. This nonlinear function can be represented as a two-dimensional cubic polynomial.

$$I_{sp} = \begin{bmatrix} MR^3 & MR^2 & MR & 1 \end{bmatrix} P_{43} \begin{bmatrix} T^2 \\ T \\ 1 \end{bmatrix} \qquad (4)$$

where I_{sp} is the specific impulse measured in N-sec/kg, MR is the dimensionless mixture ratio, T is the engine thrust measured in N, and P_{43} is a 4×3 matrix measured in sec/kg computed to be

$$P = \begin{bmatrix} -9.054\left(10^{-7}\right) & 6.155\left(10^{-3}\right) & 6.523\left(10^{1}\right) \\ 1.097\left(10^{-5}\right) & -7.571\left(10^{-2}\right) & -7.763\left(10^{2}\right) \\ -4.378\left(10^{-5}\right) & 3.069\left(10^{-1}\right) & 2.910\left(10^{3}\right) \\ 5.273\left(10^{-5}\right) & -3.734\left(10^{-1}\right) & -1.630\left(10^{2}\right) \end{bmatrix}$$

The absolute error between equation (4) and the JANNAF result was determined to be on average less than 0.12% indicating equation (4) to be a satisfactory, albeit complex, equation that relates the mixture ratio and engine thrust to the specific impulse. The "effective" specific impulse (for equivalent free-space delta-V calculations) was assumed to be 99.5% of the specific impulse determined via equation (4). This additional 0.5% reduction in performance accounts for the duty cycle and canting of the TVC thrusters on each stage.

Warm-Gas Pressurization System

Both stages of the MAS assumed a warm-gas pressurization system. This warm-gas pressurization system assumed that small amounts of hydrogen and oxygen (at stoichiometric ratios) are added to the helium pressurant mass. The gas mixture catalyzes in a pressurant reactor downstream of the pressure regulators. For cryogenic propulsion systems, the amount of pressurant required could be reduced by 60% or more by using a warm-gas pressurization system compared to a conventional pressurization system. This paper assumed a 200 K temperature rise in the propellant tank ullage during ascent. Pressurant savings of 60% and a temperature rise of over 200 K have been demonstrated in cryogenic propulsion systems.[12]

Propellant and Pressurant Tanks

Both stages of the MAS assume advanced lightweight composite overwrapped propellant tanks and lightweight composite overwrapped pressurant tanks. The propellant tanks are near spherical and assume a 0.1778 mm (7 mil) thick stainless steel liner, a 0.127 mm (5 mil) thick adhesive, and a varying thickness polybenzoxazole (PBO) low-angle composite overwrap. The pressurant tanks are cylindrical (length/diameter of 1.5) and assume a 0.1778 mm (7 mil) thick aluminum liner, a 0.127 mm (5 mil) thick adhesive, and a varying thickness T-1000 low-angle composite overwrap. The mass assumed for the propellant and pressurant tanks as a function of internal volume is shown in Figure 11.

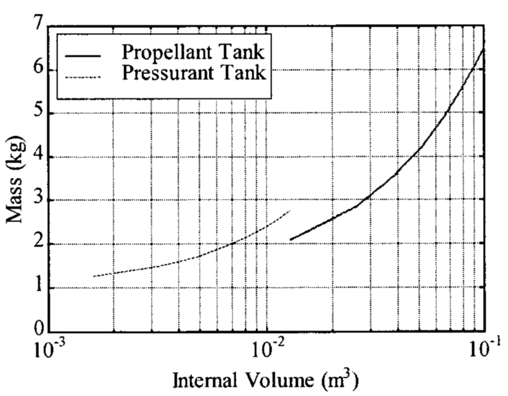

Figure 11. Mass as a Function of Internal Volume for the MAS Propellant and Pressurant Tanks.

It should be noted that Figure 11 does not include the mass of the propellant management device (PMD), contingency, or insulation. A PMD mass 0.5 kg was assumed for each propellant tank on the MAS. A 2% initial ullage was assumed. A 5% volume and 25% mass contingency were placed on all tanks. As stated earlier, additional insulation mass was applied to all propellant tanks to minimize the boil-off of propellant during ISPP. Since no credible method of jettisoning the insulation was determined without significantly sacrificing the thermal insulating ability of the insulation, the insulation mass is kept during ascent.

Propulsion Components

Both the first and second stage propulsion systems assume several lightweight components. Table 2 summarizes the masses assumed for all the propulsion components.

Table 2. Propulsion Component Masses Assumed.

Propulsion Component	Mass (kg)
Service valve	0.01
Pyrotechnic valve (NC)	0.15
Filter, gas	0.15
Regulator, gas	0.50
Pressurant reactor	0.10
Burst disk	0.15
Pressure transducer	0.18
Temperature transducer	0.01
Lines, brackets, & fittings	2.00

The quantities of each component on each stage of the MAS can be determined from Figures 7 and 8. A flat 25% dry mass margin was placed on all the propulsion components listed in Table 2.

Other Subsystems

The pseudo-guided MAS utilizes an avionics package that is simple enough to achieve a low-Mars orbit. The avionics package includes a small flight computer; thermal battery; power distribution slice; power control slice; valve and pyrotechnic drive circuitry; inertial measurement unit; sun sensor; multi-layer insulation; sensors; and structure to house these items. A mass of 7 kg was assumed for the avionics package. The mass of the sample, sample container, and sample container jettison device was estimated to be 5 kg. A 25% dry mass margin was placed on the avionics package, sample, sample container, and sample container jettison device. The mass of structure, mechanisms, and cabling on the first stage was estimated to be 5% of the entire wet mass of the MAS. The mass of the aerodynamic fairing, structure, mechanisms, and cabling on the second stage was assumed to be 5% of the wet mass of the second stage. General thermal control mass for each stage was assumed to be 1.3% of the wet mass for that stage.

Lander Assumptions

As was stated earlier in the introduction, the Lander comprises of a cruise stage, backshell, heatshield, and landed element. The Lander assumed in this study is the lander being designed for the 2003/2005 MSR missions with modifications. A different power subsystem is assumed that can provide the significant amounts of power required by the payload and the ISPP cryogenic cooling system. A different method of sample collection was also assumed via a robotic arm. Modifying the 2003/2005 lander instead of developing a custom lander for 2007/2009 will result in significant cost reduction to the overall 2007/2009 MSR mission. This section begins with a discussion of the lander power assumptions. A discussion of the lander configuration assumptions follows. The section ends with a discussion of the lander mass assumptions.

Power Assumptions

This section begins with a description of the availability of power on Mars. A short summary of the power requirements follows. The section ends with a description of each of the three distinct areas of the electrical power subsystem: power generation, energy storage, and power electronics.

Power Availability on Mars

In general, significant amounts of electrical power will be required for ISPP and cryogenic storage. This power level would have to be delivered for as many hours as possible during a 300 sol span. Since it is unlikely that nuclear power will be permitted for robotic missions, the power will have to be obtained using large extended arrays of photovoltaics. Power production then depends on latitude and season.

The ellipticity of the Mars orbit around the sun results in the solar intensity above the atmosphere being 45% greater when Mars is closest to the sun than when it is furthest from the sun. Since Mars is closest to the sun in summer in the Southern Hemisphere, the summers in the Southern Hemisphere are much warmer and the winters are colder than in the Northern Hemisphere. In the Northern Hemisphere, the differences between summer and winter are mitigated by the tilt of the axis of rotation of Mars. The solar intensity is greater in winter when the sun is lower in the sky. In general, the smallest variations of solar intensity with season over the course of a Martian year (668.6 sols) are encountered between the equator and about 15° N latitude. However, for shorter durations, the highest solar availability could be at negative latitudes in local summer.

For a perfectly clear atmosphere, the difference in solar elevation between local summer and winter manifests itself merely as a cosine effect on the irradiation of a horizontal surface. However, the turbidity of the Mars atmosphere assures that the effect of differences in solar elevation will be accentuated by the absorption and scattering that takes place along the longer path length when the solar elevation is lower in winter.

For pure absorption (no scattering) the following relationship (Beer's law) holds:

$$I_{ground} = I_{extraterrestrial} \cdot e^{-D/\cos(Z)} \qquad (5)$$

where D is the dimensionless "optical depth" and Z is the elevation angle (measured in degrees or radians from the vertical) of the sun. The optical depth is a parameter that represents the integral of the absorption coefficient over the vertical path length through the atmosphere. If absorption was the only process taking place, this would describe the fraction of extraterrestrial irradiance that reaches the Mars surface as a function of optical depth and elevation angle. However, it turns out that most of the absorption of sunlight by the Mars atmosphere occurs in the ultraviolet, and overall, absorption is only a minor factor in the passage of the full spectrum of sunlight through the Mars atmosphere. Scattering by dust is much more important. J. B. Pollack and co-workers performed a detailed analysis of scattering for an assumed type of dust particle.[13] Their results provide the total downward flux onto a horizontal surface as a fractional transmission coefficient for any dust optical depth and solar zenith angle. This transmission coefficient includes direct beam transmission plus diffuse irradiance. Using their results, the solar irradiance was computed on a horizontal surface at 15 °N latitude as a function of season (see Figure 12).

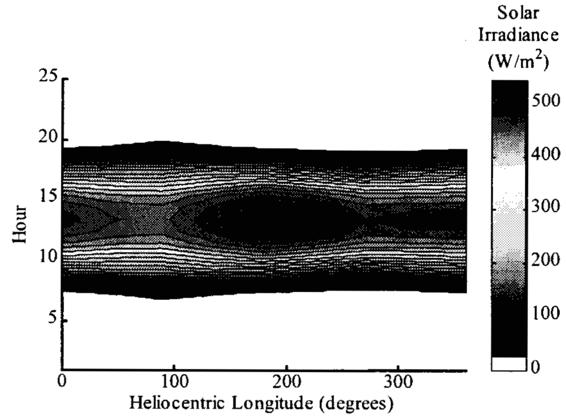

Figure 12. Solar Irradiance at 15°N Mars Lattitude for Various Seasons.

The seasons are indicated in Figure 12 by the heliocentric longitude (HL) which varies from 0° to 360° during a Martian year (with Mars being at the closest distance to the Sun near HL = 250°). It should be noted from Figure 12 that at 15° N significant solar irradiance is available only for ~7 hours during the entire Martian year. The daily total insolation on a horizontal surface as a function of heliocentric longitude and Mars latitude is shown in Figure 13.

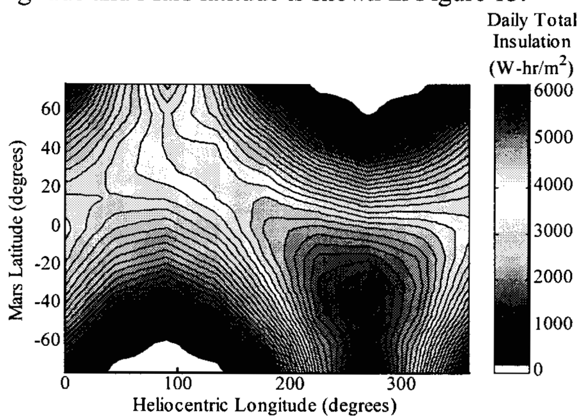

Figure 13. Daily Total Insolation on a Horizontal Surface as a Function of Heliocentric Longitude and Mars Latitude.

It can be seen that the smallest variation of insolation with season over a full Martian year occurs near 15° N latitude. However, for applications that only last ~300 sols, southern latitudes in local summer would be more favorable. For a 300 sol period starting at either a HL of 28° at 15° N latitude (2007 MSR mission) or a HL of 51° at 15° N latitude (2009 MSR mission), the daily total insolation varies from about 3600 W-hr/m^2 to about 4300 W-hr/m^2.

Power Requirements

The power requirement during the Earth – Mars cruise is dominated by the hydrogen cryogenic maintenance power (see Figure 6). In addition a constant power requirement of 50 W was assumed for general housekeeping functions. A 30% contingency was placed on the total power requirement during cruise in sizing the solar arrays.

The power requirements of the landed element over a given sol are fivefold: cryogenic maintenance of the hydrogen (day and night), zirconia system operations (day only), S/E system operations (day only), ISPP cryogenic maintenance, and general housekeeping functions (day and night). The power requirements of the first four have previously been discussed. A constant power requirement of 150 and 50 W was assumed for general housekeeping functions during the day and night, respectively. A 7 hour "day" (3.5 hours around mid sol) and 17.5 hour "night" were assumed per sol. A 30% contingency was placed on the total power required by these five items.

Power Generation

Power generation by the cruise stage is accomplished via two deployable solar arrays mounted to the cruise stage structure. The gallium arsenide solar arrays assumed are capable of providing 50 W/m^2 at Mars arrival. It should be noted that the 2003/2005 lander cruise stage design does not have nor require these two deployable solar arrays. The higher power requirement during cruise for cryogenic maintenance of the hydrogen for the 2007/2009 MSR missions necessitated additional cruise stage solar array area. Power generation by the landed element is accomplished via Ultraflex™ arrays being developed by AEC Able Engineering (Goleta, CA). An average solar flux of 430 W/m^2 on the array was assumed available for the 7 hour "day". The advanced silicon (19% efficient) arrays are capable of low intensity, low temperature performance resulting in 84 W/m^2 of power per unit area (beginning of life). This array technology has been selected for the 2001, 2003, and 2005 Mars landers.

The longevity of photovoltaic solar arrays on Mars remains a question. If the arrays are horizontal, experience from the Mars Pathfinder mission suggests that dust accumulations might initially reduce the power available by ~0.3% per day. It is likely that this initial rate of decay of performance might gradually diminish with time, but over a period 300 days, serious reductions in power might occur. It is also clear from Viking and Pathfinder pictures that dust does not adhere to vertical surfaces, so this implies that the dust is mainly loose and non-binding. Therefore, a modest tilt of the photovoltaic panels might be very effective in reducing dust accumulations. Methods for dust removal have not been investigated to any degree, but include electrostatic methods as well as use of jets of high-pressure carbon dioxide from the MAAC. Degradation due to dust accumulation, electromagnetic radiation, temperature variations, and micrometeoroid impacts over a 300 sol mission was assumed to be 19%.

Energy Storage

The power requirement for energy storage of the landed element is based on the 17.5 hour overnight power load. Lithium-ion batteries with a maximum duty cycle of 80% were assumed. This type of battery should meet the 300 cycle mission life in addition to the ~1 year Earth – Mars cruise. A single-string energy storage design was assumed with a volumetric energy density of 140 W-hr/L. This energy storage technology is expected to be flight validated on the Mars 2001 Lander. Energy storage requirements of the cruise stage are satisfied via this lithium-ion battery.

Power Electronics

The power electronics are required to regulate, distribute, and condition power for the landed element, ISPP cryogenic storage, and ISPP power. In addition battery, pyrotechnic, motor, and valve drive electronics are required. Power electronics being developed at JPL for the Advanced Deep Space Technology Program (X2000) were assumed in the single string electrical power subsystem electronics design. Significant uncertainty exists as to how the high cryogenic power and ISPP loads are switched.

Configuration

The lander assumed in this paper is the lander currently under development for the 2003/2005 MSR mission with some minor modifications. Both the 2003/2005 and 2007/2009 lander designs comply with the launch envelope requirements of the Boeing Delta III/IV-class and Lockheed-Martin Atlas III/EELV MLV-class vehicles.

Backshell, Heatshield, and Landed Element

The 2007/2009 lander uses an entry, descent and landing (EDL) system that consists of a heatshield and backshell for entry heat absorption and deceleration, a mortar-fired parachute system for descent, culminating in a powered landing on 3 deployable legs. This EDL system concept is identical to that being used on the Mars Polar Lander and Mars 2001 lander. A single monopropellant propulsion system is used for cruise, entry, and controlled descent. All lander electronics are mounted inside an insulative enclosure just below the lander top deck, isolating the systems from the harsh thermal environment outside the spacecraft. The articulated lander high gain antenna used for direct-to-earth communication is the only large lander-related hardware component on the deck. The rest of the lander deck is clear for payload accommodation. This was the major design philosophy for the 2003/2005 mission, since the ability to accommodate a wide variety of different payloads, including ascent systems, rovers, and scientific instrumentation packages, was highly desirable. In this respect, the lander could then be thought of as a kind of workhorse, in which future payloads, like an ISPP ascent vehicle demonstration, could be implemented without major modifications.

Integrating the MAS and ISPP systems outlined in this paper with the 2003/2005-workhorse lander presents three configurational challenges. The first configurational challenge is that the current workhorse lander configuration allows for ~0.8 m of headroom from the lander deck to the bottom of the parachute canister in the cruise configuration. The size and location of the propellant and pressurant tanks on the MAS will drive the headroom. Secondly, the considerable power requirement during Mars operations for an ISPP and associated cryogenic power system creates a configurational challenge in accommodating adequate solar array area. The current workhorse lander design provides 8 m^2 of solar array area by way of its two deployable tracking arrays. With the additional power requirement of the ISPP and cryogenic systems, additional array area will be required. The configuration in this study assumes that additional tracking solar arrays deployed from the lander payload deck can be added to provide the additional power required. The third configurational challenge is in accommodation of the two large hydrogen tanks needed for the ISPP system. The two hydrogen tanks are mounted directly on the lander top deck. To limit hydrogen losses during cruise and Mars operations, each tank is surrounded by several millimeters of thermal insulation.

The configuration of the lander deck assumed is shown below in Figure 14.

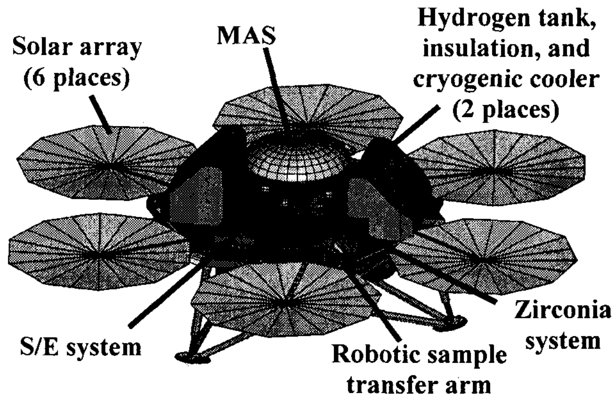

Figure 14. Lander Deck in Deployed Configuration.

As can be seen in the Figure 14, the MAS is centralized on the lander deck, surrounded by the elements of the ISPP system and the hydrogen tanks with their thermal enclosures. Also included on the deck is the sample acquisition and transfer system, which consists of a lander mounted sampling arm. The arm will obtain regolith from the area immediately surrounding the lander and deposit the material in the sample container residing at the top of the second stage of the MAS (see Figure 1). The sample container is then closed and pyrotechnically sealed to avoid back contamination of Earth. The lander high gain antenna and ISPP cryogenic cooling system are also located on the lander deck but cannot be seen in Figure 14. Six solar arrays surround the deck payload.

Cruise Stage

A cruise stage is used during the Earth – Mars leg of the mission to provide solar power and a mounting location for cruise communications equipment, as well as star trackers and sun sensors required for control and navigation. The system is inverted for launch, with the lander launch loads carried through three structural members into the backshell, then transmitted through the cruise stage into the standard 1.194 m (47") launch vehicle adapter ring. As was discussed earlier, two deployable solar arrays are attached to the cruise stage structure.

Mass Assumptions

A set of assumptions and scaling equations were used to estimate the mass of the various lander elements. These scaling equations are valid only for preliminary analysis purposes of modifying the 2003/2005 lander design. The payload mass was comprised of the MAS (dry), zirconia system, S/E system, hydrogen, hydrogen tanks/insulation/cryogenic cooling system, robotic sample transfer arm, and ISPP cryogenic system mass.

The electric power subsystem mass was comprised of the solar arrays, batteries, and electronics. The specific power density of the solar arrays on the landed element was assumed to be 50 W/kg. The battery specific energy density was assumed to be 100 W-hr/kg. The specific energy density assumed for the electronics and packaging was 75 W/kg.

The mass of landed element (m_{landed}), lander propellant (m_{prop}), heatshield ($m_{heatshield}$), and backshell ($m_{backshell}$) were assumed to be constant:

$$\begin{aligned} m_{landed} &= 570\ kg \\ m_{prop} &= 187\ kg \\ m_{heatshield} &= 171\ kg \\ m_{backshell} &= 308\ kg \end{aligned}$$

The mass of solar arrays and electronics on the cruise stage (m_{cruise_power}) measured in kg were assumed to be

$$m_{cruise\_power} = 0.045 P_{\max} \tag{6}$$

where P_{max} is the maximum power measured in W required by lander during the Earth – Mars cruise.

The mass of the cruise stage (m_{cruise}) measured in kg was assumed to be

$$m_{cruise} = 90 + m_{cruise\_arrays} \tag{7}$$

where m_{cruise_arrays} is the mass of the solar arrays on the cruise stage measured in kg.

The injected mass from Earth was assumed to be the sum of the payload, electric power subsystem, landed element, lander propellant, heatshield, backshell, and cruise stage. It should be noted that the 2003/2005 landed element design can currently accommodate a payload and power subsystem mass of 411 kg. It is unclear if the 2003/2005 landed element design will be viable for a payload and power subsystem mass greater than 411 kg. A less massive landed element is possible through a slight redesign if the payload and power subsystem mass is significantly less than 411 kg. However, such landed mass savings are unlikely to be significant and were not considered in this paper.

Results and Discussion

This section begins with a discussion of the energy and mass results. A summary of the energy and mass values for the mission elements of the optimal design follows. The section ends with a discussion of the configuration results.

Energy and Mass

The selection of the relative amounts of oxygen to be produced by the zirconia and S/E ISPP systems depends on the desired mixture ratio of the oxygen/methane propellant combination. The three most important considerations in making the choice in mixture ratio are the power requirement of the landed element, the specific impulse of the propellant combination, and the amount of hydrogen required to be brought from Earth.

Given the partition of the day and night period loads and power requirements of the landed element, an energy balance was calculated and plotted in Figure 15 for the landed element as a function of mixture ratio.

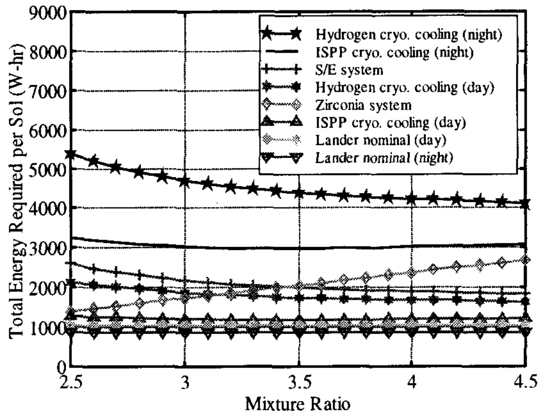

Figure 15. Total Energy Required per Sol as a Function of Mixture Ratio.

Since it is desirable to minimize the size and mass of the electronic power subsystem, Figure 15 illustrates that high oxygen/methane mixture ratios (~3.5 to ~4) minimize the overall energy requirement of the landed element. Note that the 7 hour "day" assumption is somewhat conservative since the sunlit period is longer than 7 hours. This assumption drives the size of the solar arrays and batteries. However, the sun elevation angle is low at sunrise and sunset and the resulting power from the arrays is small (see Figure 12).

The specific impulse of a propellant combination is a measure of the relative efficiency of the propellant combination compared to other propellant combinations. It is desirable to have as high a specific impulse as possible to minimize the amount of propellant that the ISPP system must produce as well as to minimize the mass of the MAS. However, it is also desirable to minimize the requirement for transporting hydrogen to Mars, thus driving the system design to very high oxygen/methane mixture ratios. Figure 16 plots the payload mass breakdown as a function of mixture ratio.

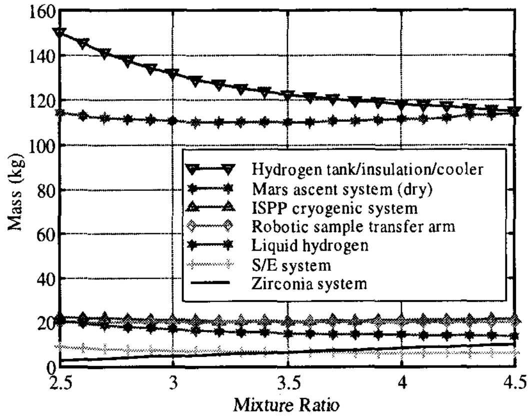

Figure 16. Payload Mass Breakdown as a Function of Mixture Ratio.

It is apparent from Figure 16 that the MAS (dry) mass is minimized at a mixture ratio of ~3.2. This value is slightly higher than the optimal mixture ratio for specific impulse of 3.1 for a wide range of thrust levels (see Figure 10). The amount of hydrogen required falls off continuously as the mixture ratio increases. At a mixture ratio of 4.0, the amount of hydrogen required is roughly 20% less than it is at a mixture ratio of 3.1. It is therefore desirable to minimize the hydrogen (and associated hydrogen tankage/insulation/cooler mass). The total payload mass is minimized at a mixture ratio of ~4.2.

Figure 17 summarizes the injected mass breakdown of the MSR mission elements as a function of mixture ratio.

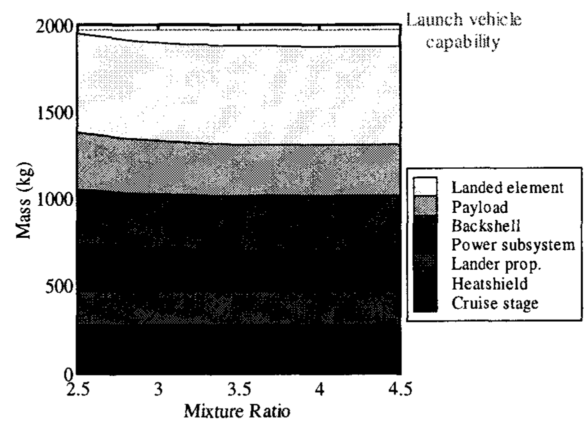

Figure 17. Injected Mass Breakdown as a Function of Mixture Ratio.

In Figure 17 is a dotted line at 1970 kg, the launch vehicle capability of the Boeing Delta IV Medium launch vehicle for a hyperbolic excess velocity of 13.25 km^2/sec^2.[14] Figure 17 also illustrates the entire range of mixture ratios between 2.5 and 4.5 satisfy the injected capability of a medium-lift launch vehicle.

Energy and Mass Values for the Optimal Design

Figure 17 illustrates that the injected mass levels off between a mixture ratio of 3.5 and 4.5. The actual injected mass is minimized at a mixture ratio of ~3.9. This, in turn, dictates that the S/E and zirconia processes should be sized to produce roughly equal amounts of oxygen. The power load required during the day and night was determined to be 1520 and 605 W, respectively. This value includes a 30% contingency on power required. The total energy required was 21220 W-hr. The energy balance that was calculated resulted in a total energy capacity for the batteries of 13227 W-hr (battery duty cycle impact included). The actual injected mass breakdown is summarized below in Table 3.

Table 3. Injected Mass Breakdown for the Optimal MSR ISPP System (Mixture Ratio = 3.9).

Item	Mass (kg)
Payload	
MAS (dry)	111.2
Zirconia system	8.0
S/E system	6.2
Hydrogen	14.4
Hydrogen tank/insulation/cooler	119.0
Robotic sample transfer arm	20.0
ISPP cyrogenic system	21.0
Electrical power subsystem	
Solar arrays	60.6
Batteries	132.3
Electronics and packaging	40.4
Lander propellant	187.0
Heatshield	171.0
Backshell	308.0
Landed element	570.0
Cruise stage	107.1
TOTAL	**1876.2**

Hence, the injected mass at a mixture ratio of 3.9 is ~1876 kg, providing a very tight 5% launch vehicle margin at this phase of the design.

Configuration

Although the energy and mass results indicate that a 2007 ISPP MSR mission is possible, a preliminary look at the resulting configuration indicates it is not possible. The six solar arrays depicted in Figure 14 are 2.5 m in diameter each providing a total solar array area of ~29 m^2. Such a ~29 m^2 solar array would provide a nominal 2478 W of power during a 7 hour day at the beginning of life (2007 W after 300 sols). However, the optimal design (at a mixture ratio of 3.9) requires 3736 W of power at the beginning of life (3026 W after 300 sols) resulting in a required solar array area of ~45 m^2. Configurational constraints of the 2003/2005 lander design prohibit arrays larger than ~29 m^2.

In fact the ~29 m^2 solar array area depicted in Figure 14 has several problems. First, it is not a practical design since only a small area of the Martian surface is available for sample collection via the robotic arm. Second, the hydrogen tank insulation is clipped by the backshell. Either a backshell mount would be required or the tanks would have to be sunk in the lander deck. Both options would require a significant reconfiguring the 2003/2005 lander design. Third, there is very little margin left in the MAS height before interference occurs with components of the lander (see Figure 2). Since the MAS design is already a low profile, removing any additional vehicle height would be difficult without reducing the ascent propellant volume. In fact, the MAS was only able to fit within the 0.8 m height limit due to the warm-gas pressurization system which reduced the size and volume of the MAS pressurant tanks considerably. Forth, the two hydrogen tanks should be nested below the top deck of the landed element to provide improved thermal insulation compared to the current design. However, due to the volume of these tanks, there is not enough space available below deck to allow for such a configuration.

It is worth noting that the center of gravity of an ISPP MSR landing "package" is significantly better compared to a solid or BYOP liquid MSR design. In a solid or BYOP liquid MAS, the entry and landing center of gravity of the entire landing package are both considerably higher than that of an ISPP system, since during the entry phase of the mission, the MAS propellant tanks are still empty. A high center of gravity during entry and landing can create stability problems that result in additional attitude control propellant requirements and increased susceptibility to tipping. These issues would not be paramount in the implementation of an ISPP-based system.

Conclusions

This paper determined that the choice in mixture ratio of the liquid oxygen and methane propellant used for ascent has a moderate impact on the overall injected mass from Earth of the MSR mission elements. Based on the assumptions made in this paper, a 2007 ISPP MSR mission satisfies the injected mass constraint of an affordable medium-lift launch vehicle yet cannot be accomplished using a modified 2003/2005 "workhorse" lander due to configuration and packaging issues. Configuration, power, thermal control, and propulsion appear to be the four areas with the highest impact on the overall feasibility and injected mass for an ISPP MSR mission.

Current solar array technology being used on the 2003/2005 MSR lander design clearly results in too large a solar array area for the 2007/2009 ISPP mission described. Technology development in landed array concentrator technology can solve much if not all of this problem. An inflatable concentration array (10+ times) would reduce the mass and volume significantly. Such an inflatable concentrator array would provide extensive solar array area while stowing in a small, lightweight volume, ideal for implementation in an already crowded payload space. However, inflatable arrays are still a developing technology and considerable design, qualification, and testing would be necessary to use them in this kind of application. The design of this concept was deemed beyond the scope of the paper.

It should be stressed that the assumptions made significantly impact the results and conclusions. Thermal requirements for the night storage of hydrogen and ISPP oxygen and methane drive the mass and volume of the Lander batteries. This electronic storage requirement in turn drives the size of the solar arrays. If the thermal assumptions made are too conservative, a reduction in the electronic power subsystem is possible. If a different configuration for the MAS and/or ISPP system were assumed, different results and conclusions would be reached. Perhaps a MAS configuration utilizing an ISPP liquid propellant first stage and the 2003/2005 MAV second and third stages could result in a system that both satisfies the injected mass constraint of a medium-lift launch vehicle and the configurational constraints of the 2003/2005 "workhorse" lander design. However, based on the results of this paper, it is unlikely that a feasible, practical, entirely ISPP-based MSR mission can be accomplished in 2007/2009 timeframe using the existing 2003/2005 lander design without technology investment.

Recommendations

Continued technology development in power generation (specifically inflatable solar arrays); power storage; cryogenic cooling; ISPP systems; advanced lightweight propellant and pressurant tanks; warm-gas pressurization systems; and low-mass propulsion components are required to enable a near-future ISPP MSR mission. A technology development schedule and cost estimate with sufficient margin needs to be devised to ensure these technologies are developed successfully, on time, and within a budget. A more detailed end-to-end design of the MSR ISPP mission elements focusing on configuration, thermal control, power, and propulsion should be performed to determine overall feasibility and further areas of required technology development.

Acknowledgments

The authors would like to thank the following for their contributions to this work: Mark Adler (Mars Exploration Directorate – JPL), Henry Awaya (Thermal & Propulsion Engineering – JPL), Wayne Lee (Mission & Systems Architecture – JPL), Sylvia Miller (Mars Exploration Directorate – JPL), Bob Miyake (Office of Mechanical Systems Group – JPL), and John C. Smith (Navigation and Flight Mechanics – JPL). The authors would also like to acknowledge the work done by Lockheed-Martin Astronautics (Denver, CO) in the design of the 2003/2005 lander assumed in this study.

The work described was performed at the Jet Propulsion Laboratory (JPL), California Institute of Technology, under contract with the National Aeronautics and Space Administration (NASA).

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